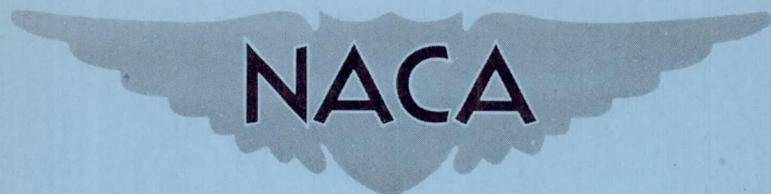


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RESEARCH MEMORANDUM

HORIZONTAL-TAIL LOAD MEASUREMENTS AT TRANSONIC SPEEDS
OF THE BELL X-1 RESEARCH AIRPLANE

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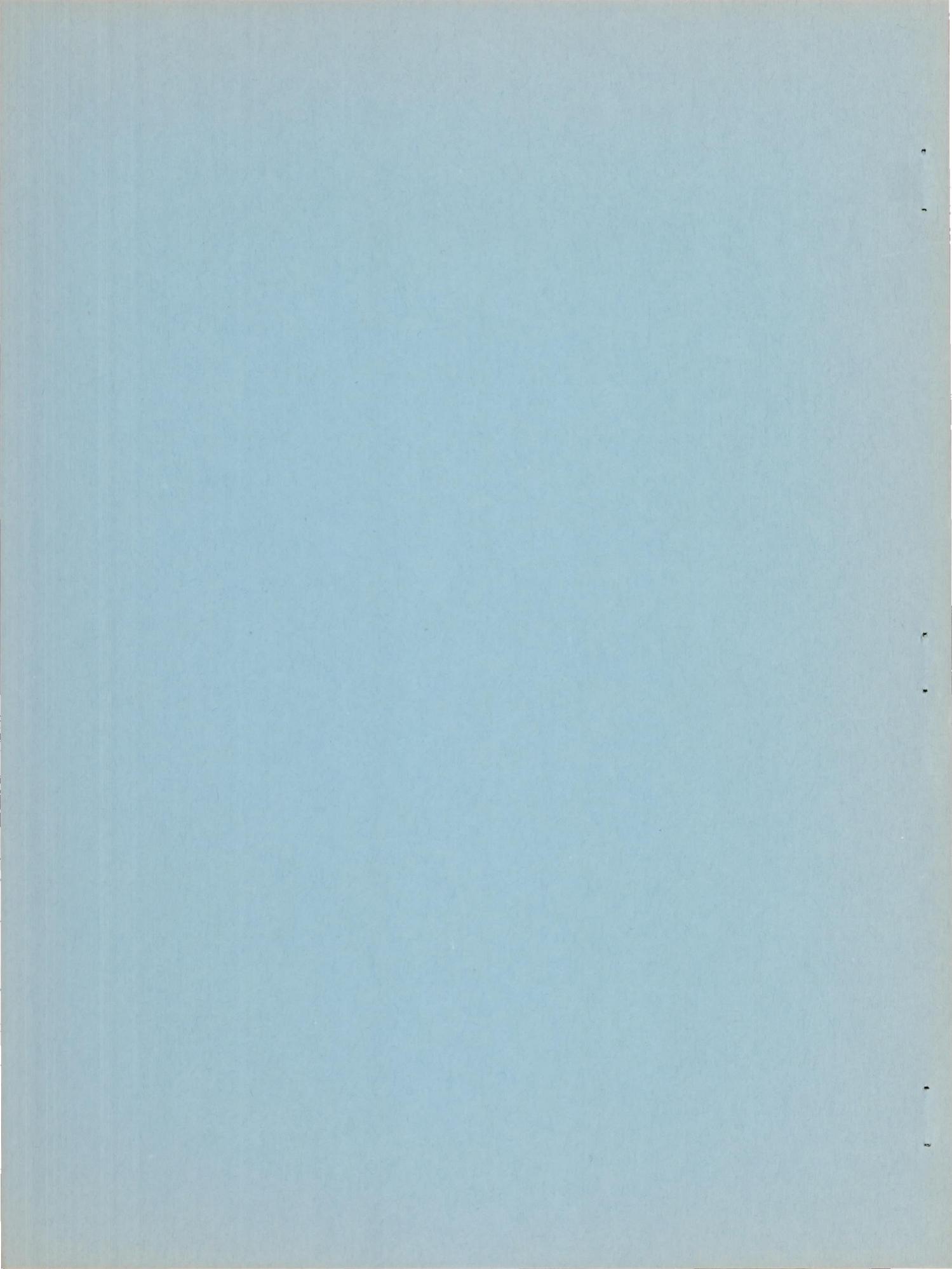
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RESEARCH MEMORANDUM

HORIZONTAL-TAIL LOAD MEASUREMENTS AT TRANSONIC SPEEDS

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SUMMARY

A flight investigation was made to determine horizontal-tail loads at transonic speeds of the Bell X-1 research airplane. The tests were made throughout the transonic region and to high lift coefficients. Comparisons between the measured flight loads and the tail loads calculated from force data of the X-1 model are presented.

For the lift range investigated the variation of tail loads with lift was linear. The loads varied with Mach number due to a rearward movement of the wing-fuselage aerodynamic center and a change in the zero-lift pitching-moment coefficient with an increase in Mach number. Comparisons between the measured tail loads and those calculated from force data of a similar wind-tunnel model indicated that for design purpose the wing-fuselage aerodynamic center could be determined satisfactorily from wind-tunnel tests. However, discrepancies were shown for the wing-fuselage zero-lift pitching-moment coefficient at the high transonic and low supersonic Mach numbers.

INTRODUCTION

A flight investigation to explore the lift-coefficient range of the Bell X-1 airplane at transonic speeds was made at the NACA High-Speed Flight Research Station at Edwards Air Force Base, Calif. The flights were made at high altitudes where possible in order to minimize the expected buffet loads and to reduce load factors required for high lift. The airplane was instrumented for the evaluation of the over-all buffeting characteristics of the airplane, the horizontal-tail loads, and the over-all drag of the airplane. The results of the over-all drag measurements are presented in reference 1. The results of measurement of the horizontal-tail loads are presented herein.

SYMBOLS

$(a.c.)_{WF}$	aerodynamic center of wing-fuselage combination, percent mean aerodynamic chord
$(C_{m0})_{WF}$	wing-fuselage zero-lift pitching-moment coefficient, $M_0/qS\bar{c}$
C_L	lift coefficient
C_{N_A}	airplane normal-force coefficient, nW/qS
C_{N_t}	tail normal-force coefficient, L_t/qS_t
\bar{c}	mean aerodynamic chord, ft
c.g.	airplane center of gravity, percent mean aerodynamic chord
$(dC_m/dC_L)_{WF}$	wing-fuselage static-longitudinal-stability parameter
g	acceleration due to gravity, 32.2 ft/sec ²
h_p	pressure altitude
I_Y	airplane moment of inertia in pitch, slug-ft ²
L_t	total aerodynamic horizontal-tail load, (up tail load positive), lb
L_{t_n}	tail load due to airplane normal inertia and weight, lb
L_{t_0}	tail load required to balance wing-fuselage zero-lift pitching moment, lb
$L_{t\ddot{\theta}}$	tail load due to airplane angular pitching acceleration, lb
l_t	tail length, (measured between airplane center of gravity and intersection of 0.25 chord line and midsemispan of horizontal tail; $l_t = 13.397$ ft for c.g. = 23.55 percent M.A.C.), ft
M	free-stream Mach number
M_0	zero-lift wing-fuselage pitching moment, ft-lb

n	airplane normal-load factor, g units
n_t	tail normal-load factor, g units
q	dynamic pressure, $\frac{1}{2}\rho V^2$, lb/sq ft
S	wing area, sq ft
S_t	horizontal tail area, sq ft
t	time, sec
V	free-stream velocity, ft/sec
W	airplane gross weight, lb
x	distance from aerodynamic center of wing-fuselage combination to airplane center of gravity, (positive if (a.c.) _{WF} is forward of c.g.), ft
$\dot{\theta}$	pitching angular velocity, radians/sec
$\ddot{\theta}$	pitching acceleration, radians/sec ² (positive for airplane pitching nose up)
ρ	mass density of air, slugs/ft ³

DESCRIPTION OF THE AIRPLANE

The Bell X-1 is a single-place straight-wing rocket-propelled research airplane. The airplane used in this investigation incorporated a wing and tail having a thickness ratio of 0.08 and 0.06, respectively. The stabilizer is adjustable in flight having a rate of movement of approximately 2° per second. A photograph of the airplane is given in figure 1 and a three-view drawing of the airplane is shown in figure 2. A detailed description of the airplane is given in table I.

INSTRUMENTATION

Standard NACA recording instruments were installed in the airplane to measure the following quantities:

Airspeed
 Altitude
 Normal, longitudinal, and transverse accelerations at the center of gravity of the airplane
 Pitching and rolling velocities
 Pitching acceleration
 Stabilizer and elevator positions

Airspeed and altitude were measured from the pitot-static head located forward of the fuselage (see fig. 2).

For the purpose of checking measured angular pitching accelerations and for applying inertia corrections to the measured tail loads, accelerations of the tail were also determined from an accelerometer having high response characteristics.

Response of strain gages located at the tail root sections (see fig. 2) were recorded on a multichannel recording oscillograph.

ACCURACY

The estimated accuracy of the measured quantities used in evaluating the tail loads are as follows:

<u>Quantity</u>	<u>Accuracy</u>
Mach number	± 0.01
Normal acceleration at the center of gravity, g units	± 0.01
Tail normal acceleration, g units	± 0.02
Angular acceleration, radians/sec ²	± 0.04
Tail shear, lb	± 50
Tail bending moment, in-lb	± 1500

ANALYSIS OF DATA

As illustrated in figure 3 the total aerodynamic horizontal-tail load during maneuvering flight may be considered to consist of three components:

the tail load required to balance the wing-fuselage zero-lift pitching moment, the tail load due to the airplane normal inertia and weight, and the tail load due to angular pitching acceleration of the airplane. The total aerodynamic tail load may be expressed as

$$L_t = L_{t_0} + L_{t_n} + L_{t\ddot{\theta}} \quad (1)$$

where

$$L_{t_0} = \frac{(C_{m_0})_{WF} q S \bar{c}}{l_t + x}$$

$$L_{t_n} = \frac{n W x}{l_t + x}$$

and

$$L_{t\ddot{\theta}} = - \frac{I_Y \ddot{\theta}}{l_t + x}$$

If the pitching acceleration is equal to zero or if the measured tail loads are corrected to zero pitching acceleration, the total tail load is equal to

$$L_t = (C_{m_0})_{WF} \frac{q S \bar{c}}{l_t + x} + \frac{n W x}{l_t + x} \quad (2)$$

and the tail load per g (dL_t/dn), the wing-fuselage combination static-longitudinal-stability parameter (dC_m/dC_L)_{WF}, the wing-fuselage aerodynamic center (a.c.)_{WF}, and the zero-lift wing-fuselage pitching-moment coefficient $(C_{m_0})_{WF}$ may be determined from

$$\frac{dL_t}{dn} = \frac{W \frac{x}{\bar{c}}}{\frac{x}{\bar{c}} + \frac{l_t}{\bar{c}}} \quad (3)$$

$$\left(\frac{dC_m}{dC_L} \right)_{WF} = \frac{x}{\bar{c}} = \frac{\frac{l_t}{\bar{c}} \frac{dL_t}{dn}}{W - \frac{dL_t}{dn}} \quad (4)$$

$$(a.c.)_{WF} = c.g. - \frac{x}{\bar{c}} \quad (5)$$

and

$$(C_m)_0 = \frac{L_t(l_t + x)}{qS\bar{c}} - \frac{nWx}{qS\bar{c}} \quad (6)$$

TESTS

The tests were conducted at altitudes from 14,000 to 50,000 feet and covered the lift-coefficient range to near maximum lift and over the Mach number range from 0.7 to 1.3. The data presented in this paper were obtained during level flight through the speed range under various conditions of power, center of gravity, and weight, and during power-off maneuvering flight throughout the speed range in an empty weight condition. The majority of the maneuvering flight data were obtained with the elevator fixed and the airplane maneuvered by use of the stabilizer.

RESULTS AND DISCUSSION

Figure 4 shows time histories of airplane normal-force coefficient, pitching angular velocity, and aerodynamic tail loads during typical pull-ups at subsonic, transonic, and supersonic Mach numbers. The tail-load data include the effects of pitching acceleration. Therefore, the variation of the tail load with airplane normal-force coefficient has been determined by correcting the measured tail-load data for pitching acceleration by the term $\frac{I_y \ddot{\theta}}{l_t + x}$ using a value of $I_y = 12,350$ slug-feet² obtained from oscillation tests on the ground. Typical variations are shown in figure 5. These data show that the tail load increases in upload with an increase in airplane normal-force coefficient at a Mach number of 0.70, shows little or no increase at a Mach number of 0.91, and increases in down load at a Mach number of 1.0. The change in the load variations

illustrated for these Mach numbers represents an increase in the longitudinal stability of the wing-fuselage combination as the Mach number is increased from 0.70 to 1.0.

The parameter, tail load per g , was determined by taking slopes of data of figure 5 and is presented in figure 6. By using the slopes and equations (4) and (5) the longitudinal stability parameter $(dC_m/dC_L)_{WF}$, and the wing-fuselage aerodynamic center were calculated. The variation of calculated parameters with Mach number is shown in figure 6. It may be seen that between Mach numbers of 0.70 to 0.88 the tail load per g (c.g. = 23.55 percent M.A.C.) remains approximately constant at about 150 pounds per g and in the vicinity of $M = 0.90$ decreases rapidly to a value of about -300 pounds per g where it remains essentially constant to the highest Mach number tested. The variation in tail load per g results from a rearward movement of the aerodynamic center of the wing-fuselage combination as the Mach number increases. From Mach numbers of 0.70 to 0.88 the wing-fuselage aerodynamic center is located at approximately 19 percent of the mean aerodynamic chord and as the Mach number is increased through $M \approx 0.90$ the aerodynamic center moves abruptly rearward to a position of approximately 35 percent of the mean aerodynamic chord. For a center-of-gravity position of 23.55 percent of the mean aerodynamic chord the aerodynamic-center location produced an unstable wing-fuselage combination, $\left(\frac{dC_m}{dC_L}\right)_{WF} \approx 0.05$ for the subsonic Mach numbers, produced neutral stability at Mach numbers in the vicinity of 0.90, and caused the wing-fuselage combination to become stable at supersonic Mach numbers.

Horizontal-tail-load data were obtained in straight and level flight from a Mach number of 0.7 to 1.3 at an altitude of about 40,000 feet and were corrected for variations in power, center of gravity, weight, and airplane normal-force coefficient, and are presented in figure 7 for a weight and center of gravity corresponding to an empty weight condition, a power-off condition, and for an airplane normal-force coefficient of 0.3. The value of airplane normal-force coefficient of 0.3 corresponds to an approximate mean value of airplane normal-force coefficient during level-flight tests of the X-1 airplane at an altitude of about 40,000 feet. As may be seen from the figure, there is a slightly greater down load existing at supersonic speeds than occurred at subsonic Mach numbers with irregular variations between Mach numbers of 0.85 to 0.94. The changes in the tail load result from a movement of the aerodynamic center and a change in the zero-lift pitching-moment coefficient of the wing-fuselage combination with Mach number.

Values of aerodynamic-center positions determined from the maneuvering-flight data have been used to calculate the wing-fuselage zero-lift pitching-moment coefficient from the level-flight data of figure 7. The results are shown in figure 8 as a variation with Mach number. Also included are zero-lift pitching-moment data obtained by extrapolation of the maneuvering-flight

data to zero lift. The variation of the parameter shows a change from a value of about -0.03 at a Mach number of 0.7 to approximately zero at supersonic Mach numbers with abrupt changes occurring at transonic speeds.

Generally, during the design stage of an aircraft, sufficient aerodynamic and geometric characteristics of the airplane are known so that the horizontal-tail loads may be calculated for assigned values of load factors and pitching accelerations. The aerodynamic data would usually be obtained from wind-tunnel tests of scaled models. A comparison was made of the measured tail loads and tail loads calculated by using available wind-tunnel results from a scaled model of the X-1 airplane. The purpose of this comparison was to indicate the agreement to be expected of the measured loads and the calculated loads computed from aerodynamic data and geometric characteristics which could be obtainable during the design stages of an aircraft. For this comparison flight measured load factors and pitching accelerations were used in the calculation. The comparisons between the measured and calculated loads are presented in figure 9 as time histories for Mach numbers of 0.70, 0.91, and 1.00 at altitudes of about 14,000, 33,000, and 48,000 feet, respectively. The tail loads are given in pounds. Discrepancies may be noted between the measured and calculated loads for each Mach number given. The reasons for the discrepancies will be discussed first and then the importance of the discrepancies shown will be discussed.

The wind-tunnel data used to calculate the tail loads were obtained from the Langley high-speed 7- by 10-foot tunnel by the transonic-bump technique and are reported in reference 2. These data are shown in figure 10 along with the flight data presented previously in figures 6 and 8. It may be pointed out that the methods used to determine the flight and wind-tunnel parameters differ somewhat; that is, the flight values are determined from variations of forces with angle of attack and the wind-tunnel values are determined from forces acting at a selected angle of attack. The locations of the wing-fuselage aerodynamic center at the lower Mach numbers and the magnitude of the rearward shift in aerodynamic center occurring at the higher Mach numbers for the two tests agree very well; however, there is a discrepancy in the Mach number at which the abrupt shift in aerodynamic center occurs. Comparison of the zero-lift pitching-moment coefficient shows that the trends of the two sets of data are somewhat similar but differ in absolute magnitude.

The discrepancies in loads shown in figure 9 at Mach numbers of 0.70 and 1.00 are due primarily to discrepancies in measurement of the wing-fuselage zero-lift pitching-moment coefficient shown in figure 10. The load discrepancies shown at a Mach number of 0.91 are primarily due to differences in the determination of the wing-fuselage aerodynamic-center location in a region where the aerodynamic center is moving abruptly rearward. For the X-1 airplane, under the flight conditions shown, the load discrepancies are not considered to be large. However, for a specific

aircraft design the discrepancies might be important. For instance, discrepancy in the determination of the Mach number where the rapid change in the aerodynamic center occurs would not be considered serious since the airplane would normally be designed for conditions throughout this Mach number range. However, differences in the magnitude of the zero-lift pitching-moment coefficient throughout a large range of Mach number, particularly in the case of a large aircraft, might be serious.

CONCLUSIONS

Measurements of the horizontal-tail loads of the Bell X-1 research airplane have shown:

1. The tail load per g varies with Mach number as a result of a rearward movement of the aerodynamic center of the wing-fuselage combination. The variation is essentially from an up tail load per g at the subsonic Mach numbers (Mach numbers less than 0.9) to a down tail load per g at the higher Mach numbers (Mach numbers greater than 0.9).
2. For the lift range investigated the variation of tail load with lift was linear.
3. For an airplane normal-force coefficient of 0.3, which corresponds to a mean value of airplane normal-force coefficient during level flight at an altitude of 40,000 feet, the balancing tail loads increase in a down direction as the Mach number is increased from 0.7 to 1.3 with irregular variations near a Mach number of 0.9. The changes in the tail load result from a movement of the aerodynamic center and a change in the zero-lift pitching-moment coefficient of the wing-fuselage combination with Mach number. The zero-lift pitching-moment coefficient changes from a value of about -0.03 at a Mach number of 0.7 to approximately zero at supersonic Mach numbers with abrupt changes occurring near a Mach number of 0.9.
4. Comparisons between the flight measured tail loads and those calculated from force data of a similar wind-tunnel model showed that for design purposes the wing-fuselage aerodynamic center could be determined satisfactorily from results of force data. Discrepancies were

shown, however, for the determination of zero-lift pitching-moment coefficient at the high transonic and low supersonic Mach numbers.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., June 16, 1953.

REFERENCES

1. Carman, L. Robert, and Carden, John R.: Lift and Drag Coefficients for the Bell X-1 Airplane (8-Percent-Thick Wing) in Power-off Transonic Flight. NACA RM L51E08, 1951.
2. Lockwood, Vernard E.: Aerodynamic and Lateral-Control Characteristics of a $\frac{1}{28}$ -Scale Model of the Bell X-1 Airplane Wing-Fuselage Combination. Transonic-Bump Method. NACA RM L50C22, 1950.

TABLE I

DETAILED DESCRIPTION OF THE BELL X-1 AIRPLANE

Airplane:

Weight during tests

Landing condition (no fuel), lb	7,340
Launching condition (full fuel), lb	12,400
Center-of-gravity position, percent M.A.C.	
Landing condition (no fuel)	23.55
Launching condition (full fuel)	21.95
Horizontal distance from airplane center of gravity to tail quarter-chord station (c.g. at 23.55 percent M.A.C.), ft.	13.397
Measured moment of inertia in pitch about an axis through the c.g. (I_y), slug-ft ²	12,350

Power plant:

Type	Reaction Motors, Inc., Model 6000c4
Number of cylinders	4
Average measured static thrust (each cylinder 2,300 ft pressure altitude), lb	1,500
Inclination of thrust axis relative to fuselage reference line, deg	-2

Wing:

Area, (including section through fuselage), sq ft	130
Span, ft	28
Airfoil section	modified NACA 65-108 (a=1)
Thickness (percent wing local chord)	8
Aspect ratio	6
Taper ratio	2:1
Mean aerodynamic chord, in.	57.71
Wing incidence, deg	2.5
Geometric twist, (washout root to tip), deg	1
Sweepback (leading edge), deg	5.05
Dihedral, deg	0

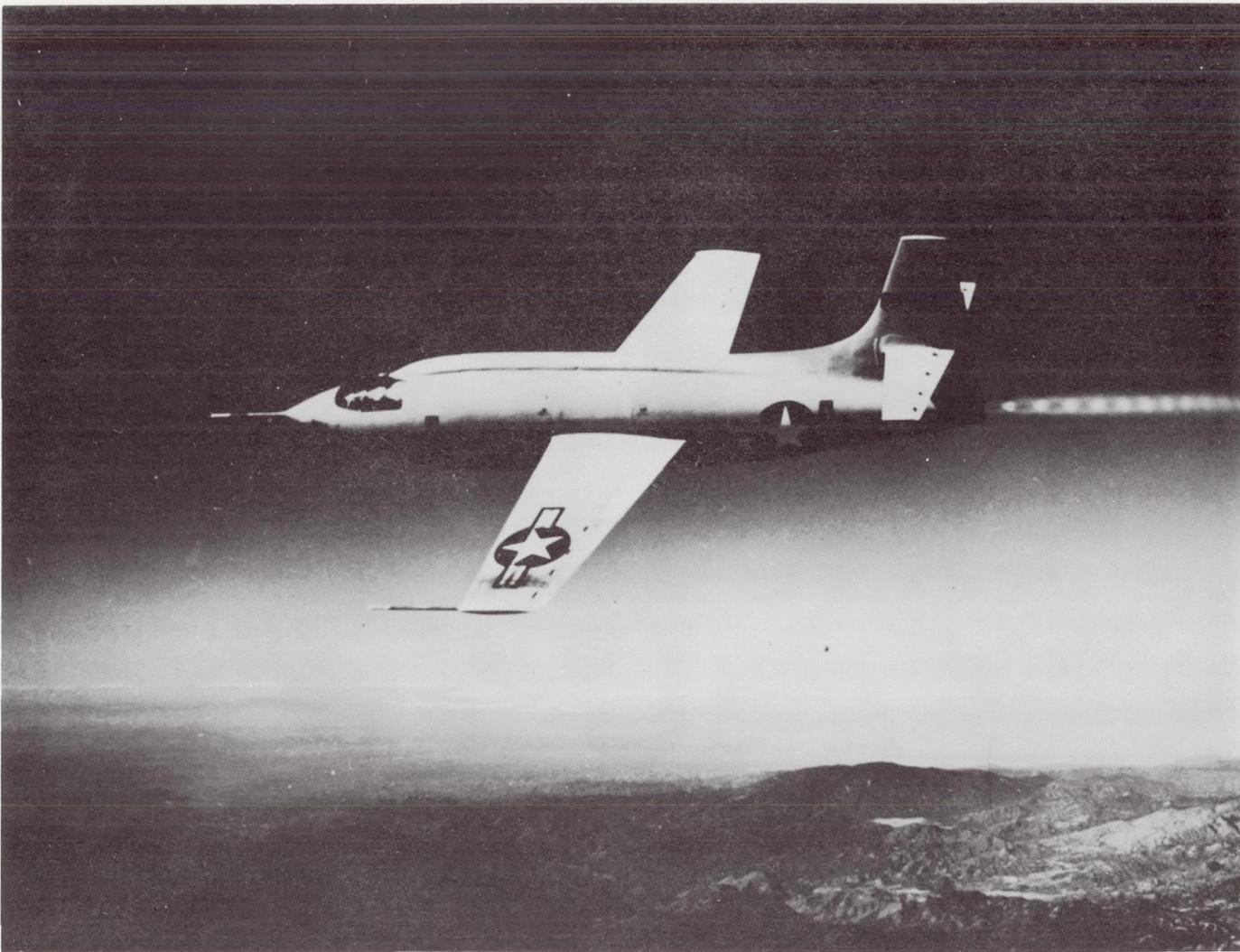
Horizontal tail:

Area, sq ft	26.0
Thickness, percent local chord	6.0
Span, ft	11.4
Aspect ratio	5.0

Elevator:

Area, sq ft	5.2
Chord, percent horizontal-tail chord	20
Approximate travel limits relative to stabilizer, deg	
Up	13
Down	11





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Figure 1.- Photograph of Bell X-1 airplane in powered flight.

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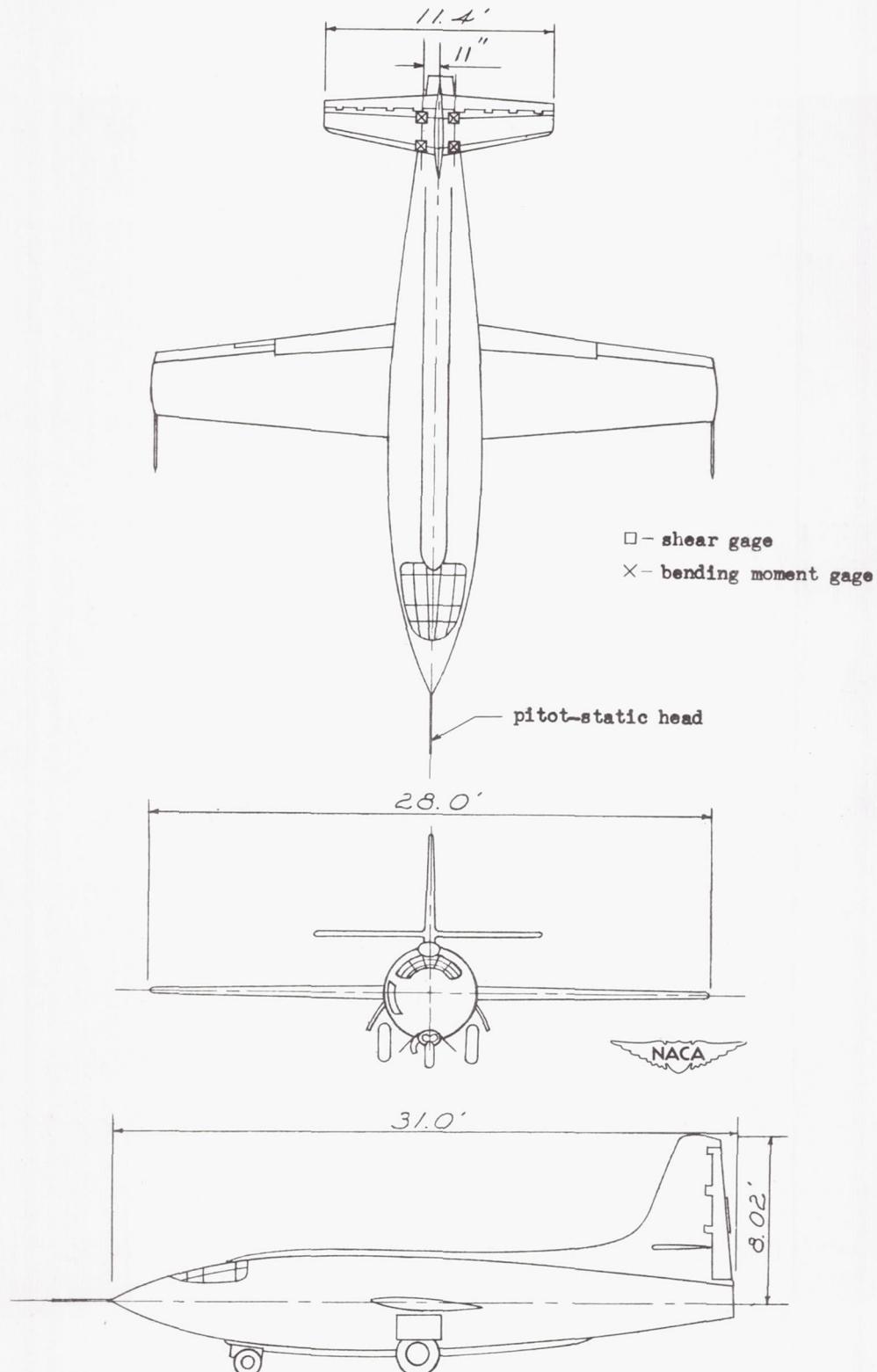


Figure 2.- Three-view drawing of the Bell X-1 airplane.

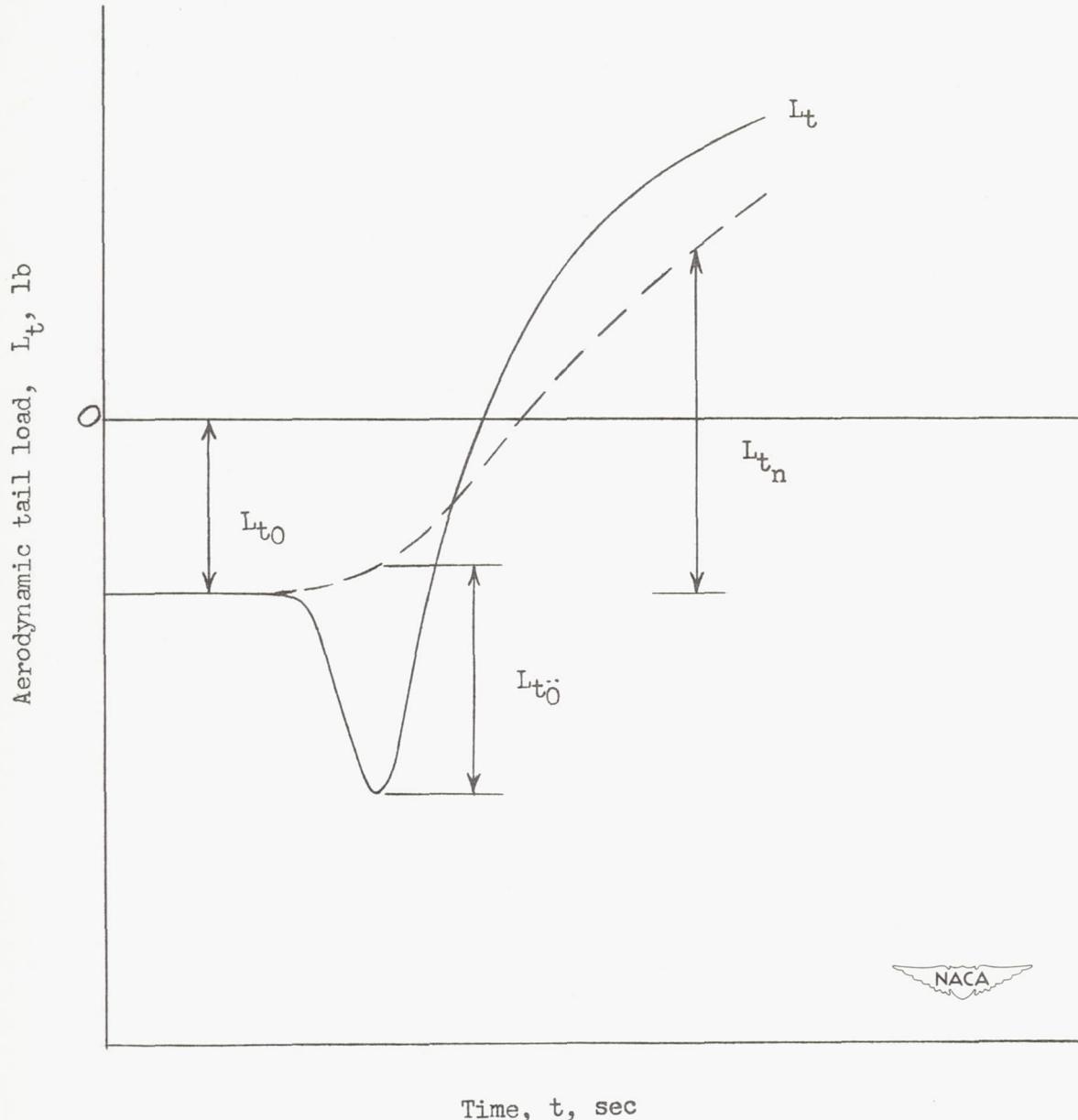


Figure 3.- Components of total aerodynamic tail load during maneuvering flight.

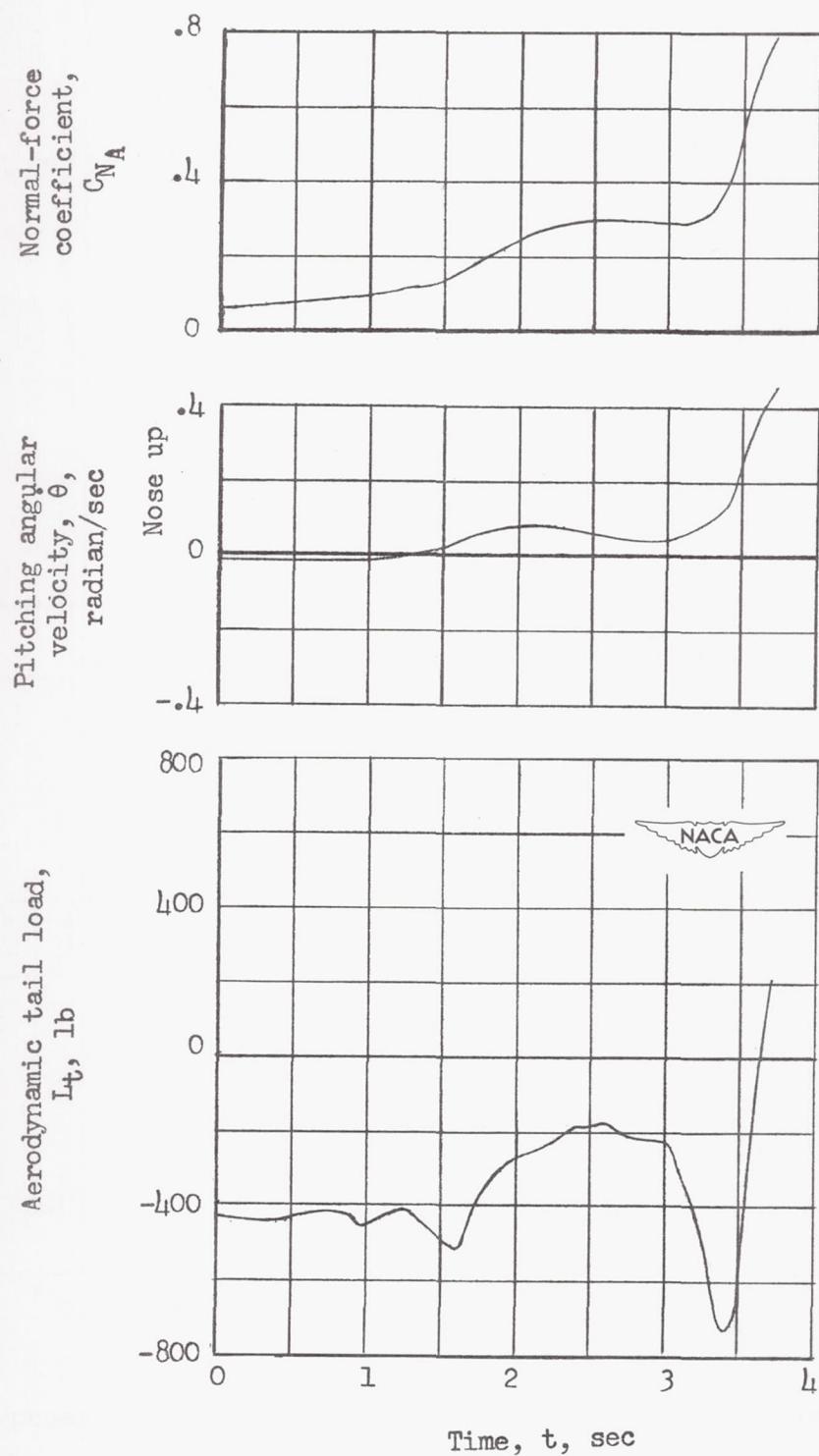
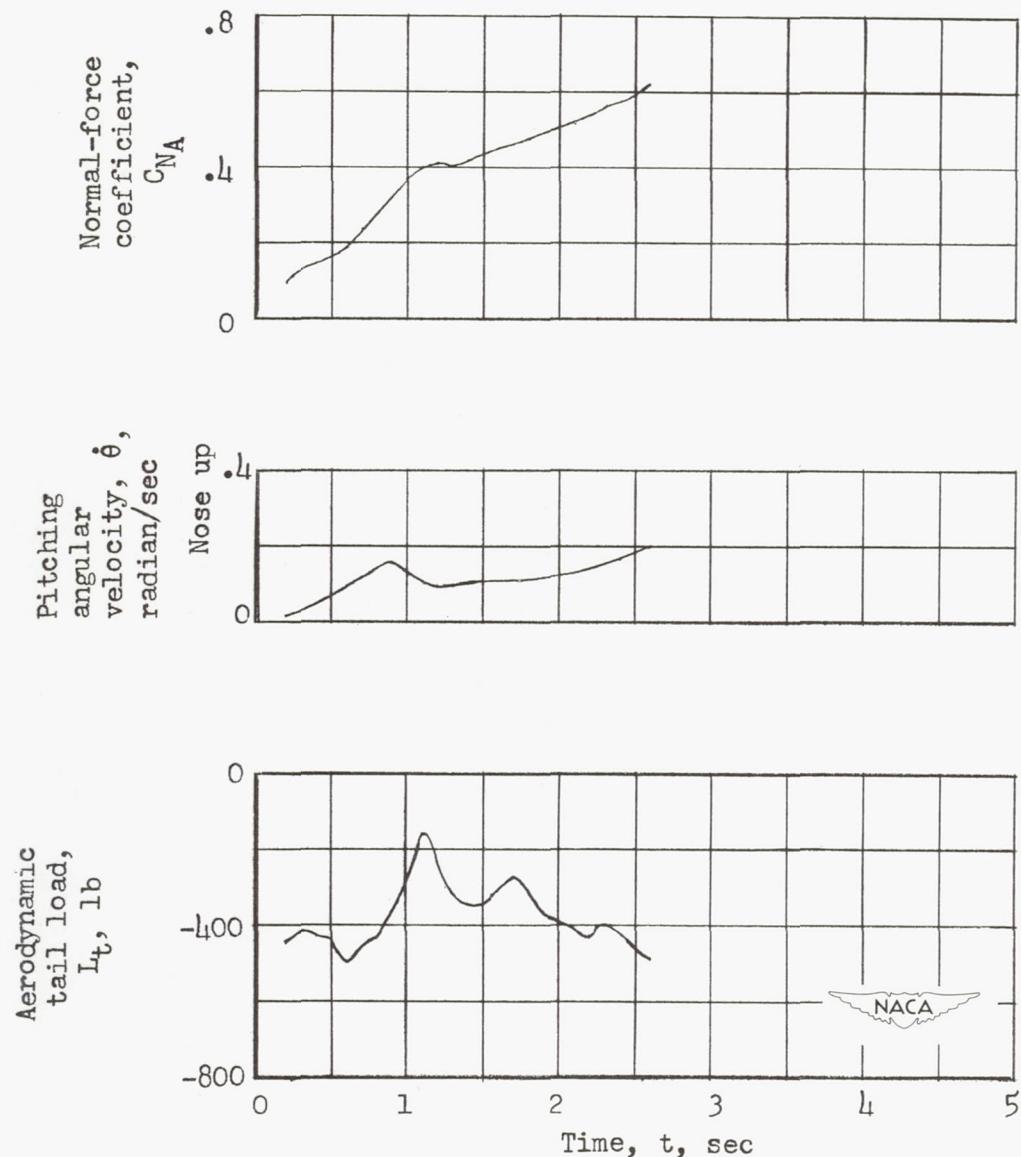
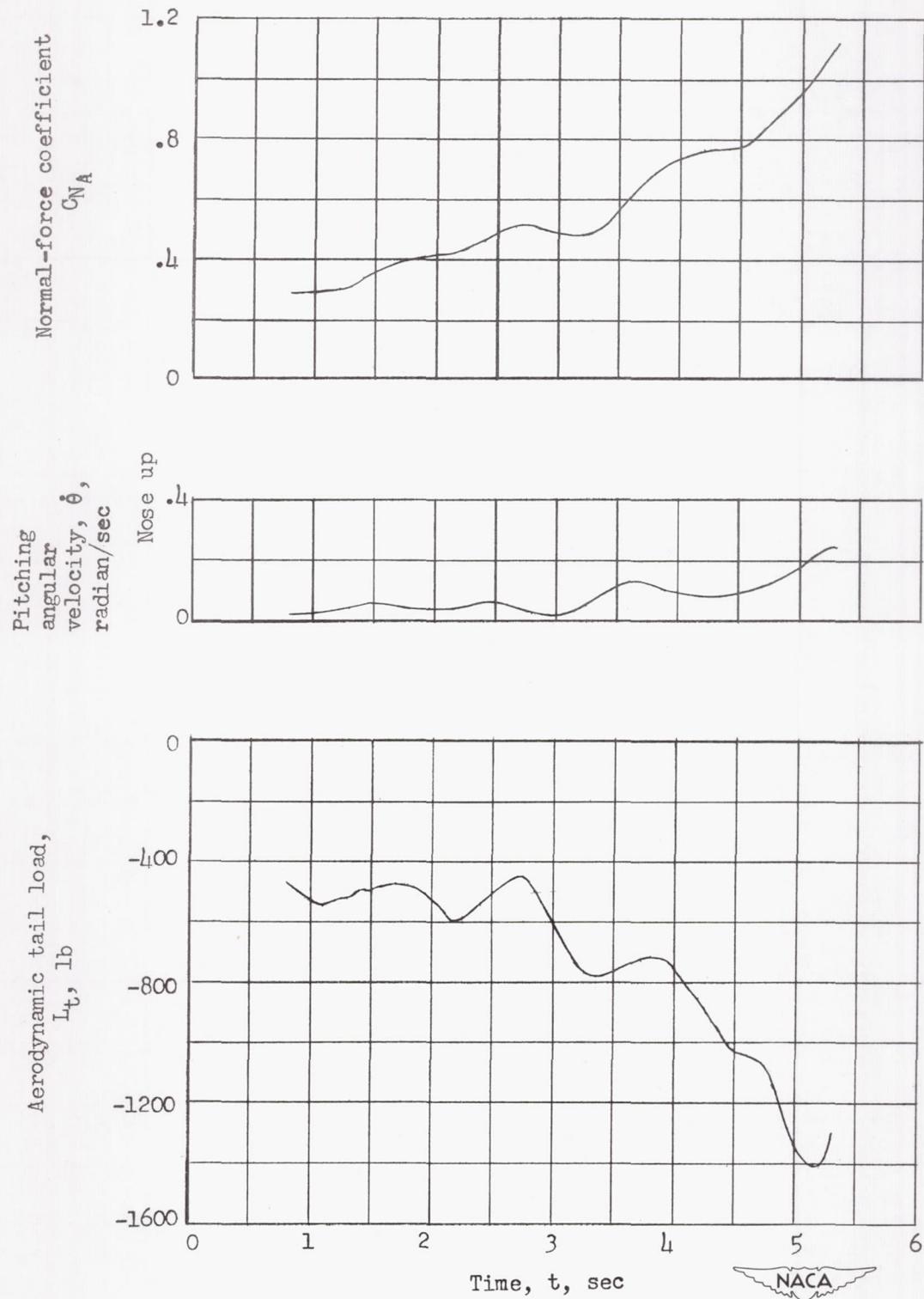
(a) Subsonic, $M \approx 0.70$; $h_p \approx 14,000$ feet.

Figure 4.- Typical time history of a pull-up for evaluation of tail loads.
Bell X-1 airplane.



(b) Transonic, $M \approx 0.91$; $h_p \approx 33,000$ feet.

Figure 4.- Continued.



(c) Supersonic, $M \approx 1.00$; $h_p \approx 48,000$ feet.

Figure 4.- Concluded.

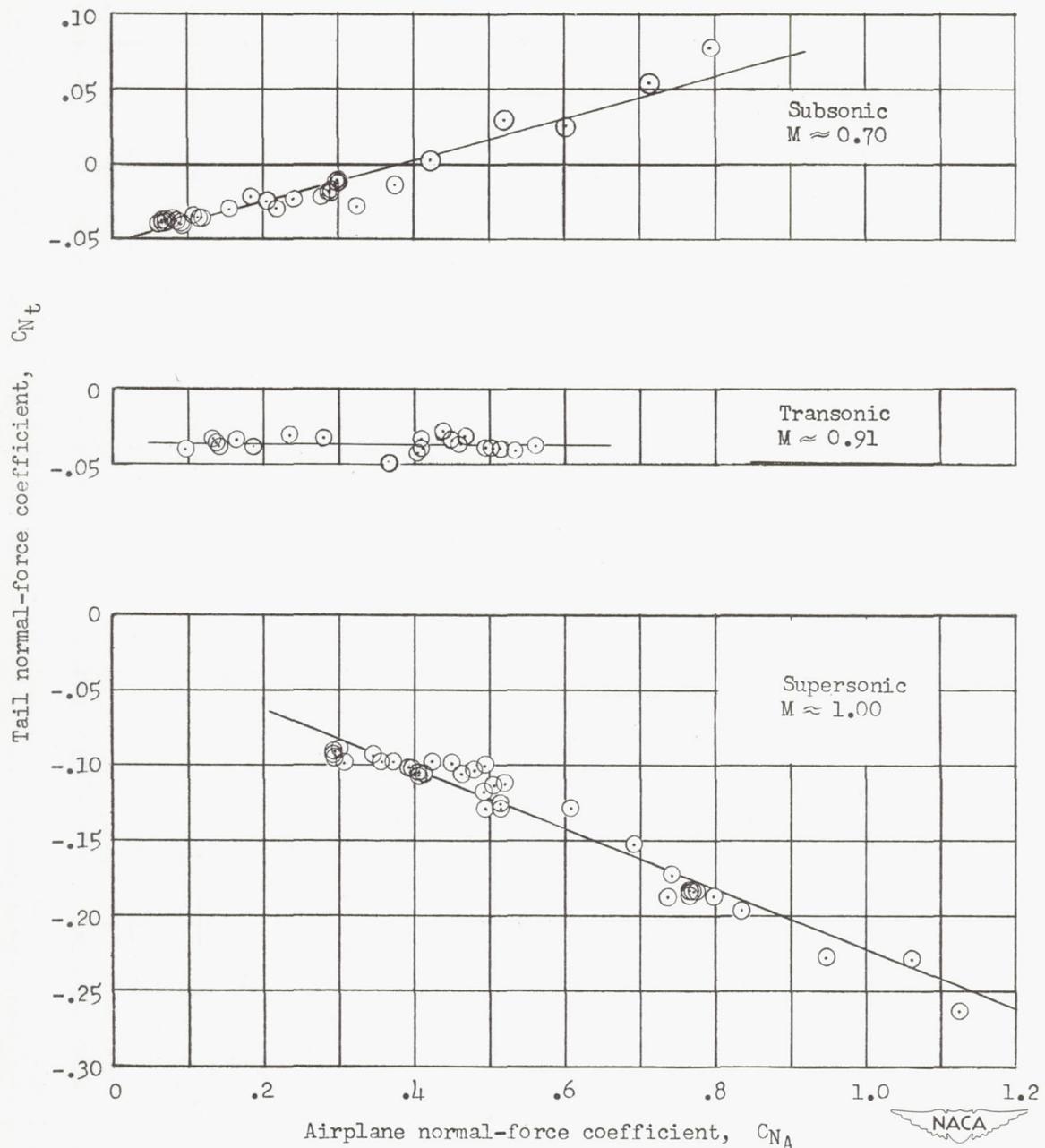


Figure 5.- Typical variation of tail normal-force coefficient with airplane normal-force coefficient during pull-ups; center of gravity at 23.55 percent mean aerodynamic chord. Bell X-1 airplane.

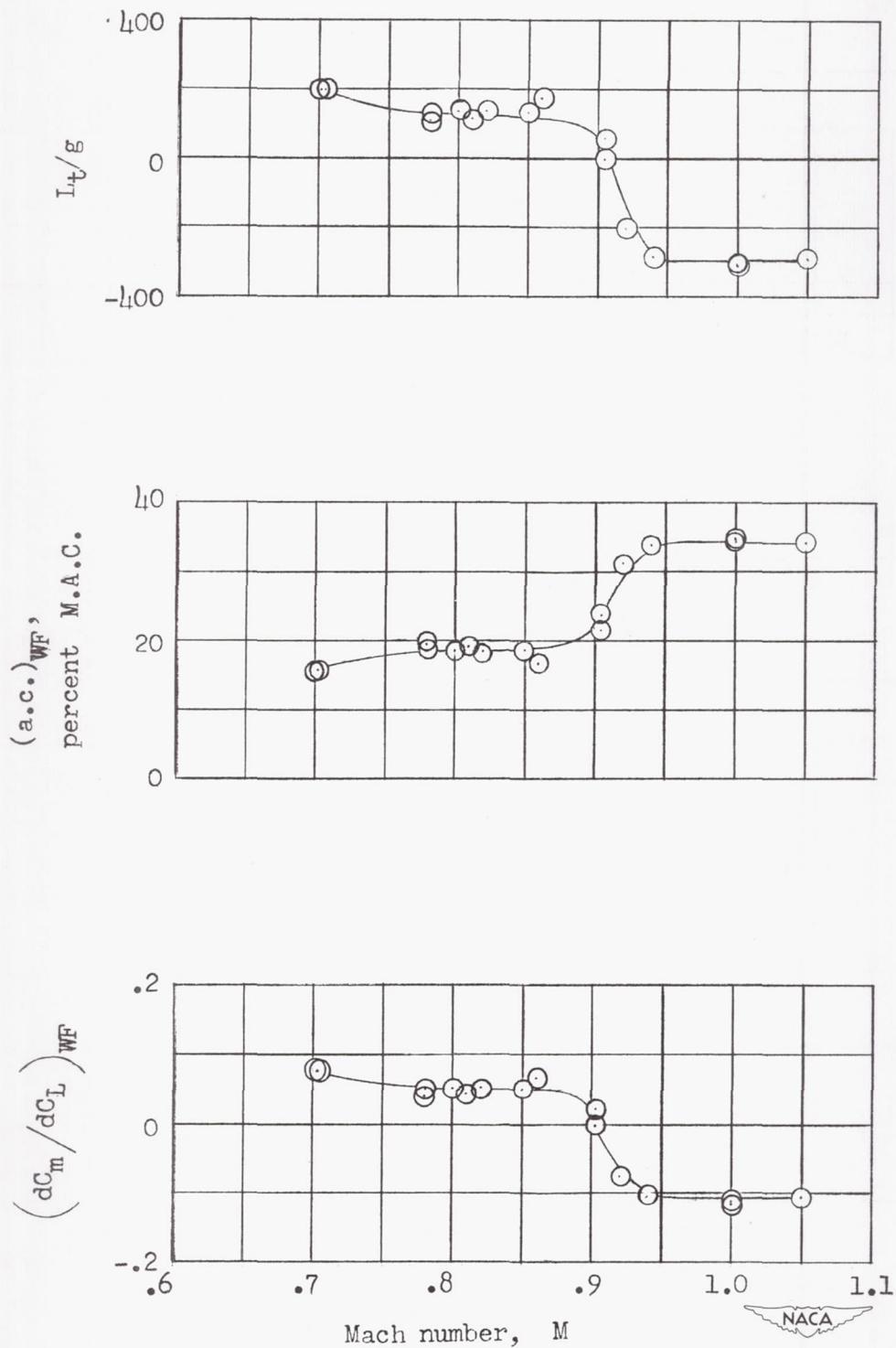


Figure 6.- Variation of tail load per g, wing-fuselage aerodynamic center, and $(dC_m/dC_L)_{WF}$ with Mach number; center of gravity at 23.55 percent mean aerodynamic chord. Bell X-1 airplane.

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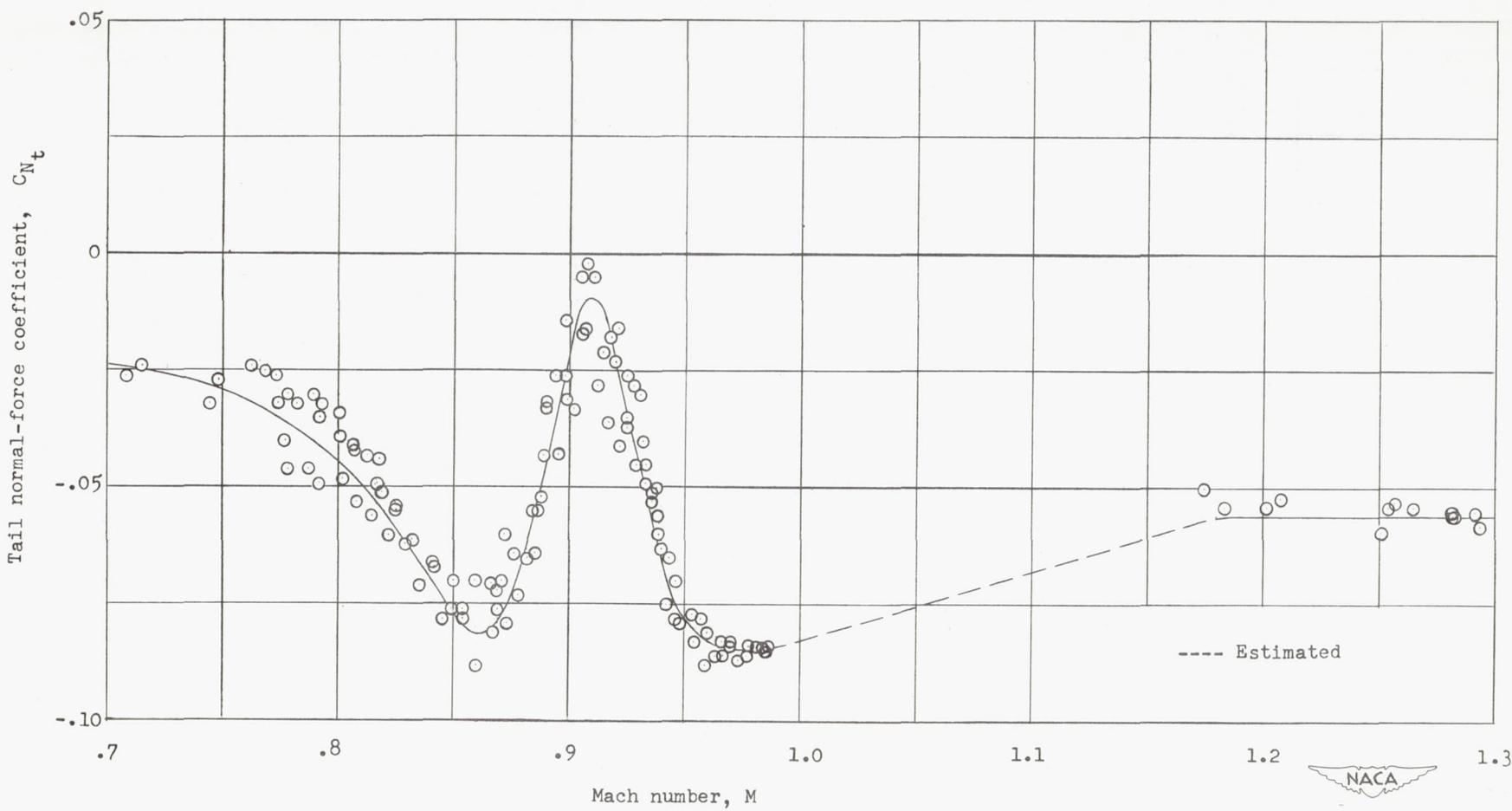


Figure 7.- Balancing tail loads for an empty weight condition and an airplane normal-force coefficient of 0.3. Bell X-1 airplane.

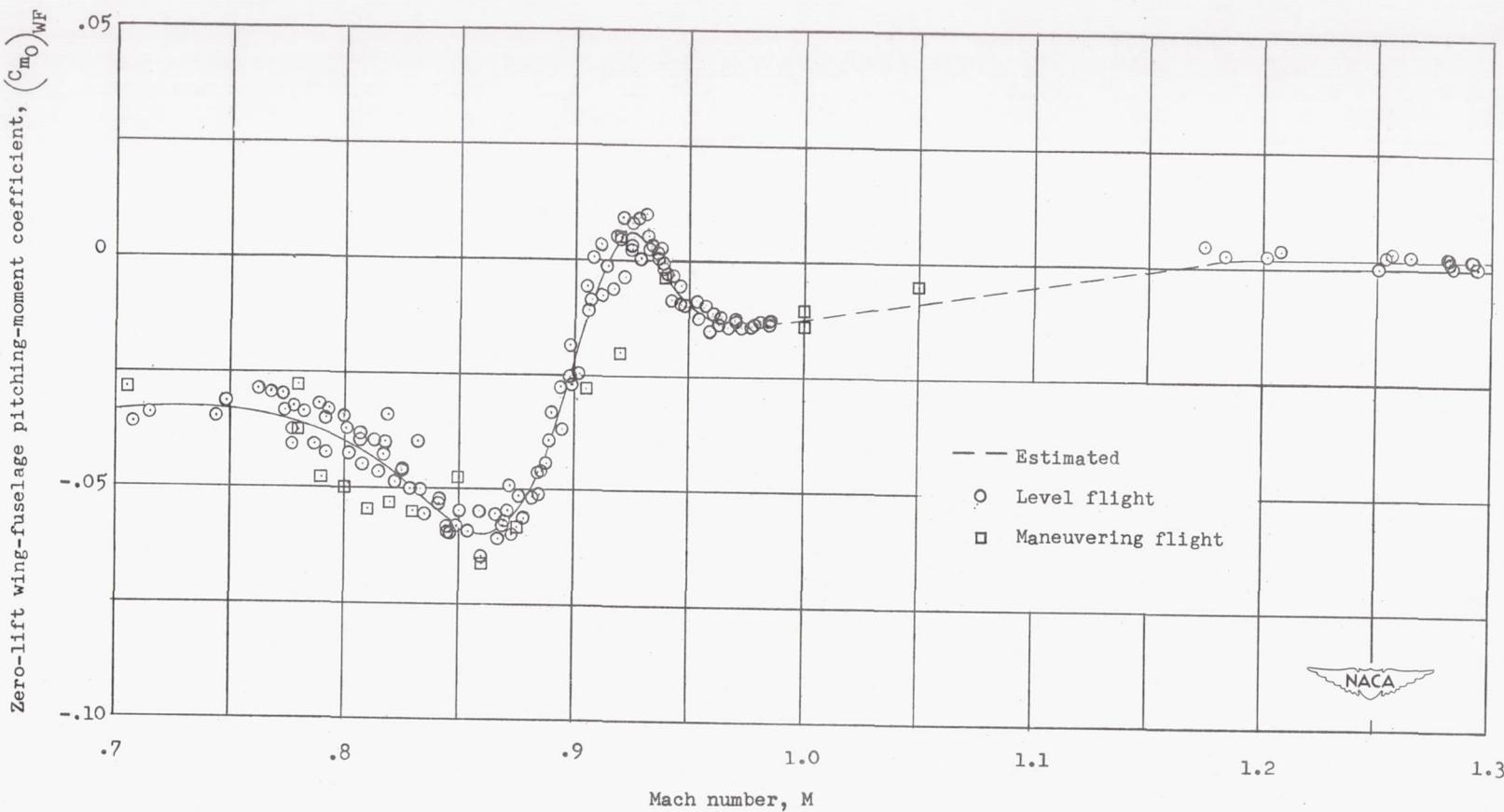


Figure 8.- Variation of zero-lift wing-fuselage pitching-moment coefficient with Mach number. Bell X-1 airplane.

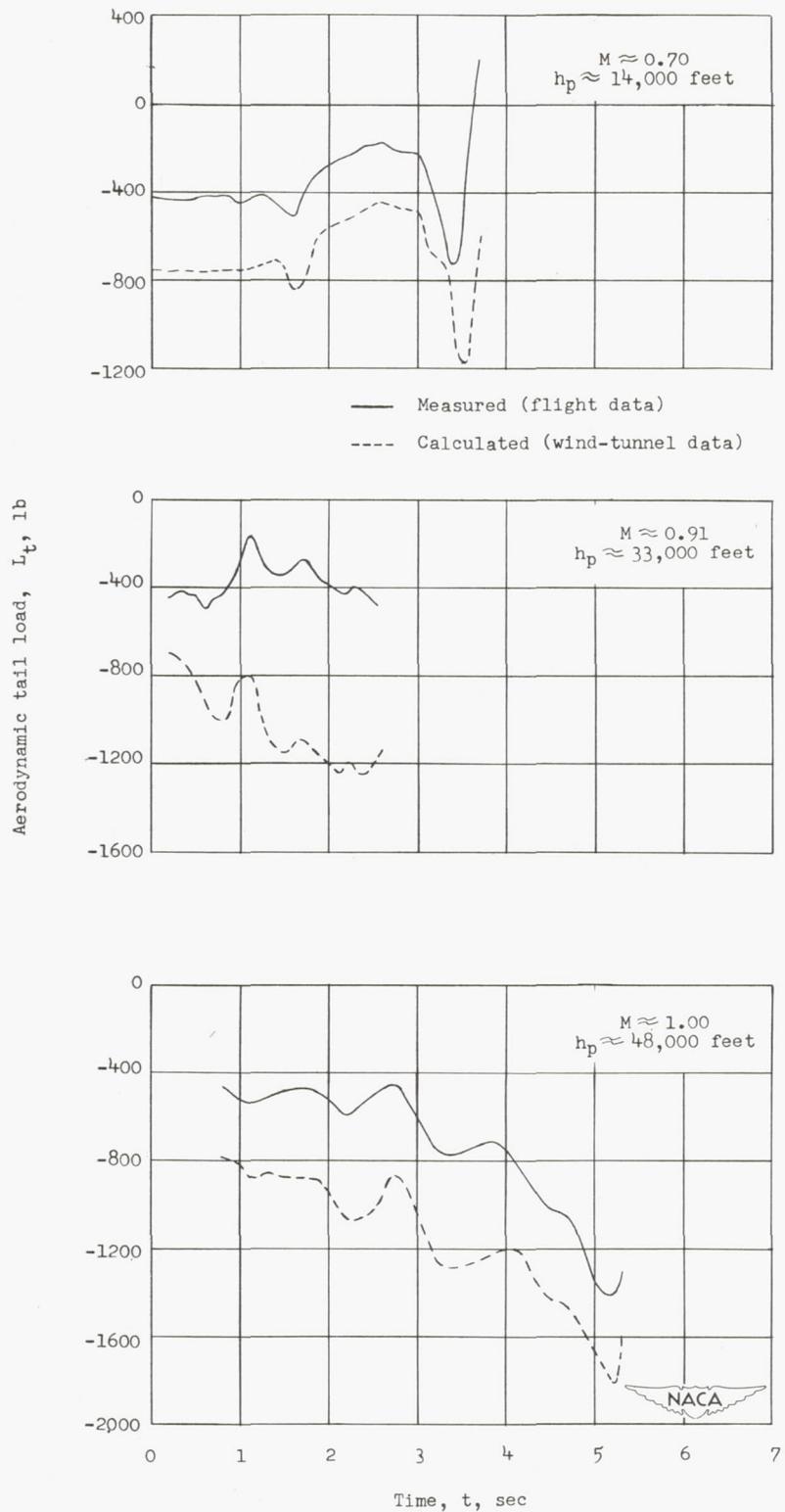


Figure 9.- Comparison of measured and calculated tail loads during maneuvering flight. Bell X-1 airplane.

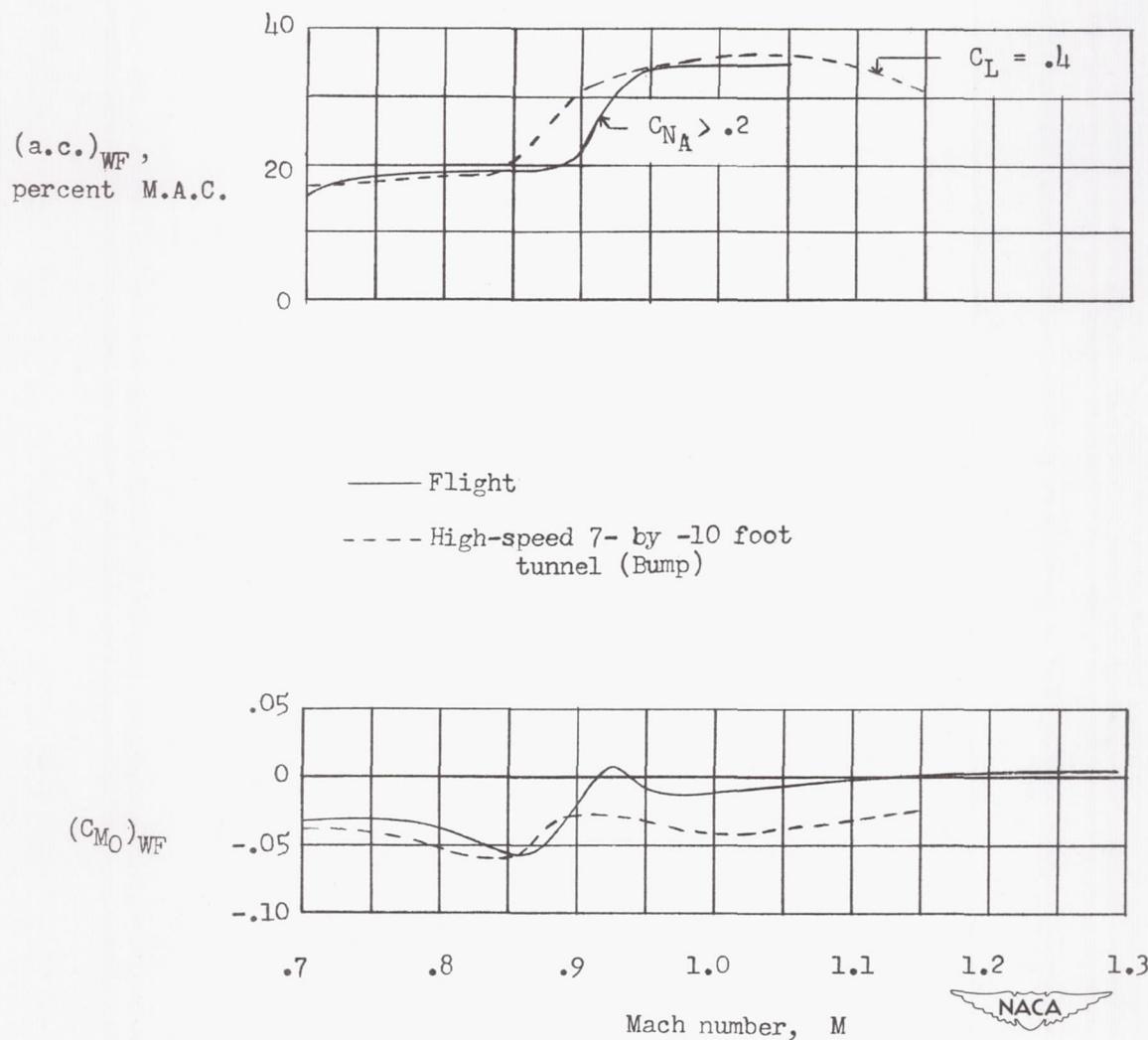
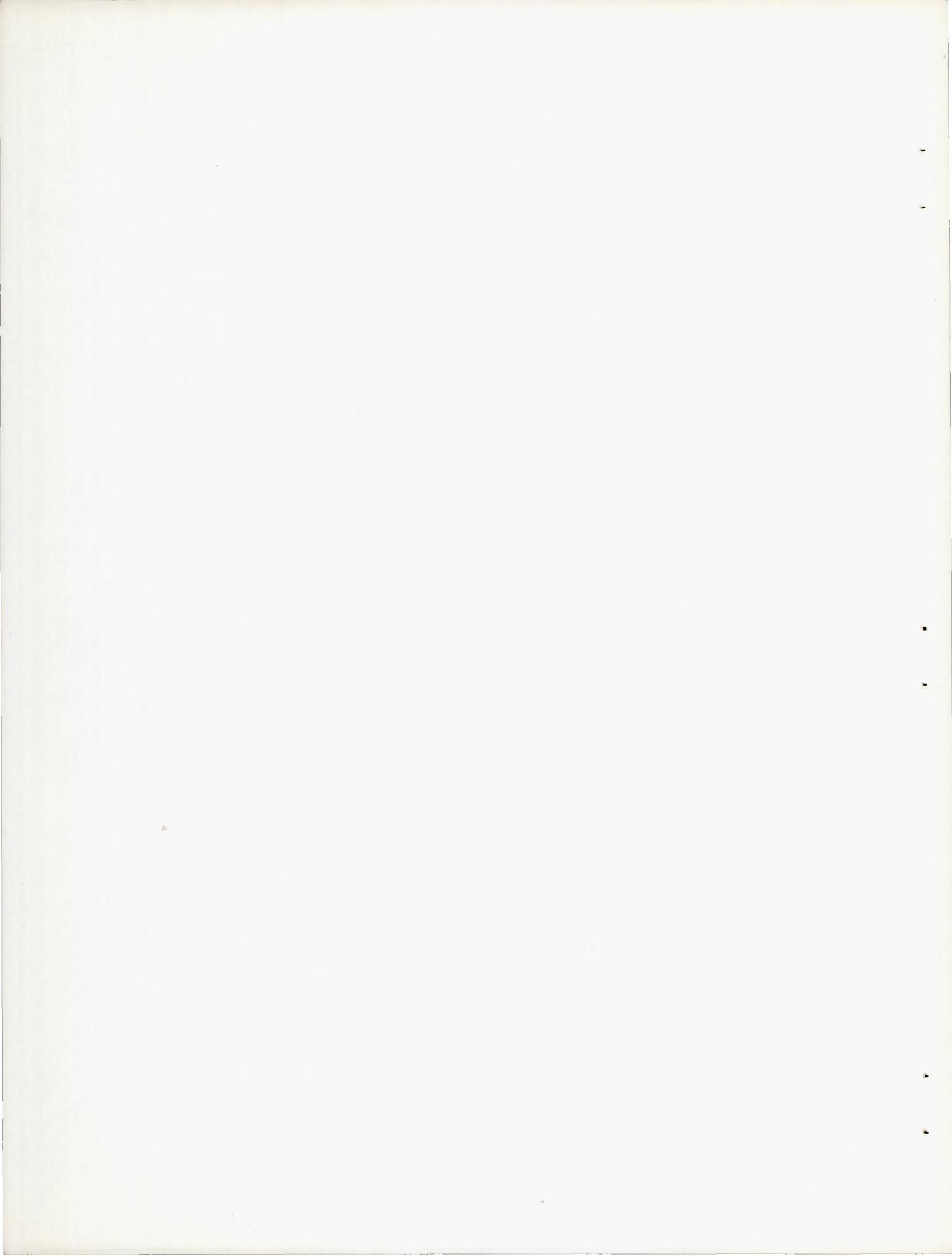
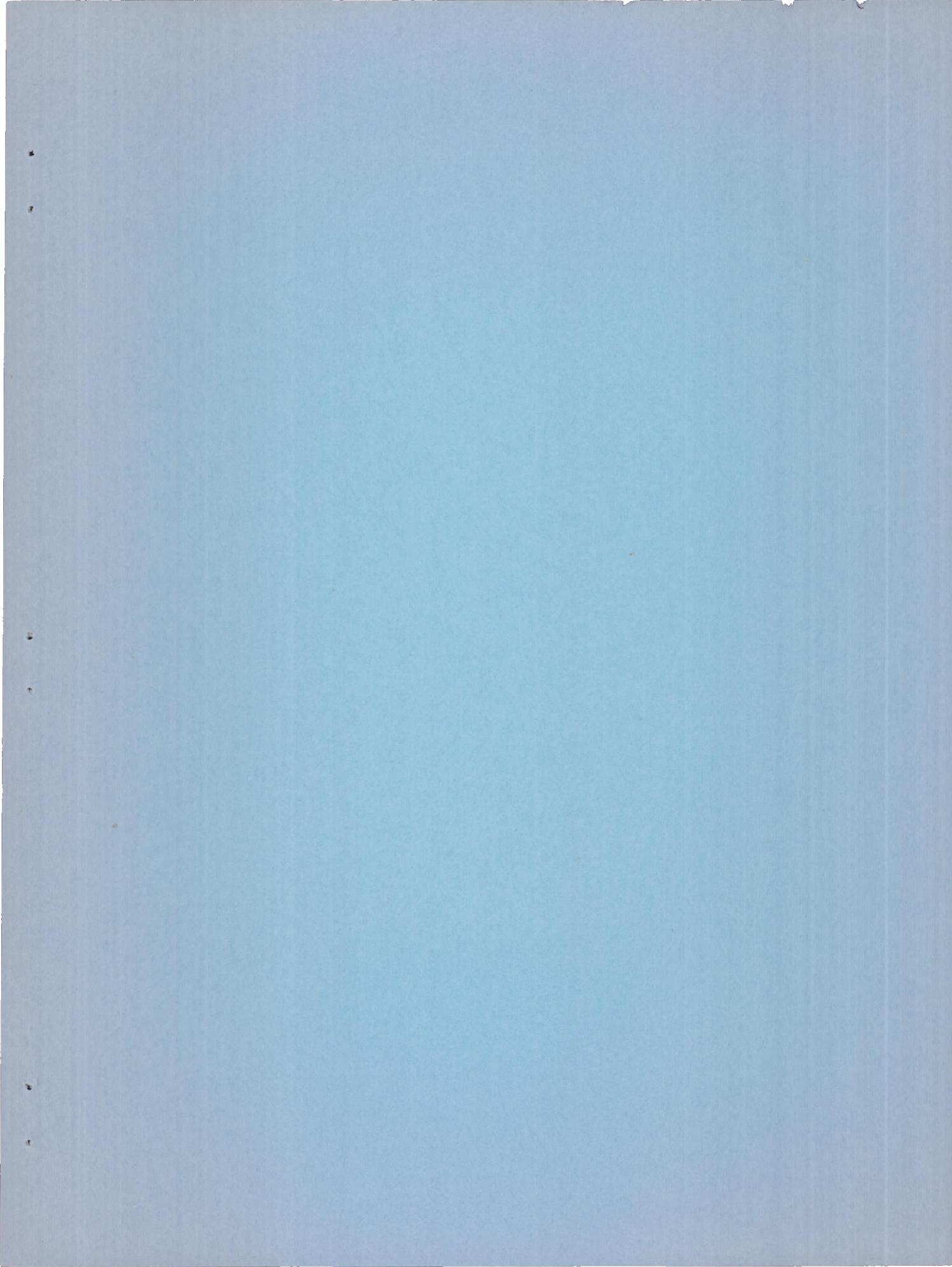


Figure 10.- Comparison of the variation of the aerodynamic center and zero-lift pitching-moment coefficient of the wing-fuselage combination with Mach number as determined from flight and wind-tunnel tests. Bell X-1 airplane.





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